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A PARAMETRIC STUDY OF CONSTANT THRUST, ELECTRICALLY PROPELLED MARS AND VENUS ORBITING PROBES

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SUMMARY

A study has been made to determine the effect of mission travel time and the vehicle performance parameters on payload for the Mars and Venus orbiting probe missions. The vehicle is assumed representative of early electrically propelled spacecraft that operate at constant thrust and constant effective specific impulse. For both the Mars and Venus missions, propellant fraction is given for a wide range of effective jet power to initial weight ratio P_{jeff}/W_0, effective specific impulse I_{eff}, and total travel time T_t. Propellant fractions are comparable for Venus missions 25 days shorter than Mars missions, and the effect of initial orbit altitude on propellant fraction is insignificant. The effects of P_{jeff}/W_0, I_{eff}, and T_t on payload fraction w_L are illustrated. In addition, the effects of specific powerplant weight α and a structural factor on maximum payload fraction are given for the range of travel times.

At maximum w_L, both α' ($\alpha' \equiv \alpha/\eta$ where η is overall thrustor efficiency) and T_t have equally significant effects on w_L; roughly the same w_L can be obtained with $\alpha' = 10$ or 30 pounds per kilowatt if T_t is allowed to increase 100 days for the Mars or Venus missions. The associated optimum values of P_{jeff}/W_0 and I_{eff} are also given for the maximum w_L cases. It is shown that both parameters are primarily affected by α' with only a slight effect due to T_t. For maximum payload fraction at low powerplant weights, the P_{jeff}/W_0 and I_{eff} should be high. Increases in the travel time require decreasing P_{jeff}/W_0 and increasing I_{eff}. The structural factor in all cases has little effect on the optimum w_L, P_{jeff}/W_0, and I_{eff}.

An example of the use of the data and the effect of a variable efficiency function (e.g., $\eta = \eta(I_{eff})$) are given for a Mars spacecraft using mercury electron-bombardment thrustors. The major effect of thrustor inefficiency on payload fraction is shown to be increased powerplant fraction because the resulting optimum propellant fraction is near the optimum value for $\eta = 1.0$. Finally, a decrease in thrustor efficiency also has the overall effect of decreasing the optimum P_{jeff}/W_0 and I_{eff}. The problem of maximum absolute payload is also discussed, and an example is given for a Mars spacecraft with a 300-kilowatt electric powerplant at $\alpha = 10$ pounds per kilowatt. Specifically, payload is maximized with respect to gross weight at fixed power.

INTRODUCTION

Electric propulsion systems are attractive for many space missions because high specific impulse and low propellant flow rate give low propellant fraction. Unlike chemical and nuclear rockets, electric rockets have high power-generation equipment weights, which may result in small payload fractions. Thus, it is necessary to carefully balance the powerplant and propellant weights for maximum payload. This balance results in low installed power causing very low thrust to weight ratios and long engine operating times. This departure from impulsive conditions demands optimization of thrust magnitude and direction to give least propellant consumption. With the appropriate set of constraints, such optimum thrust programs can be attained through the use of variational calculus. Examples are the power-limited variable-thrust program (refs. 1 and 2) and the constant-thrust program (refs. 3 and 4). The variable-thrust program is of interest because it gives the best possible performance; however, it may be unachievable for early applications because of the wide range over which thrust and specific impulse must be varied. Therefore, for early applications, the constant-thrust program is of interest. When the constant-thrust program is assumed, the effects of initial thrust to weight ratio and specific impulse on propellant fraction must be investigated. For any low-thrust mission, the total travel time is also an important parameter because of its effect on propellant requirements and mission reliability.

Several studies (e.g., refs. 4 and 5) have been made that express a payload fraction as a function of total travel time; however, they do not cover a complete spectrum of the corresponding vehicle performance parameters. In this report, a study has been made to determine the effects of both mission time and vehicle performance parameters on payload for constant-thrust Mars and Venus orbiting probe missions.

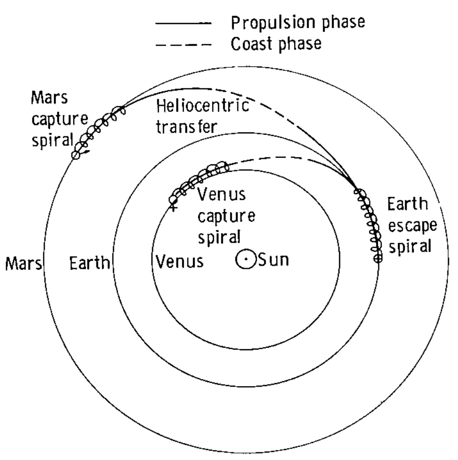
In reference 5, a study was made for the Mars orbiter mission and the Venus capture (rendezvous) mission. This study used both the variable-thrust and constant-thrust programs. When the constant-thrust program was used, it was assumed that minimization of $\int_0^T a^2\, dT$ at a given specific impulse approximates the case of minimum propellant fraction. (All symbols are defined in appendix A.) The integral itself is a parameter that arises from the constraint of constant power. Although the approximation has been shown (ref. 4) to be quite accurate (1 to 2 percent), only minimum propellant fraction data is given. Therefore, the effects of thrustor efficiencies on payload cannot be accurately assessed. Reference 5 does, however, illustrate the effects of the ellipticity of the Mars orbit by giving the minimum $\int_0^T a^2\, dT$ terminal mass for the best and worst encounters of Mars. The results indicate a difference of about 5 percent or less between the best and worst encounters.

In reference 6, payload optimization techniques were discussed in detail for the Mars rendezvous mission (heliocentric transfer only) and results were presented for transfers with optimum travel angle and best encounter of Mars elliptic orbit. Effects of both specific powerplant weight and efficiency on vehicle parameters were also discussed.

The present study assumes that the thrust and specific impulse are held constant throughout the flight, except that the engine is shut down whenever a coasting period is beneficial. During heliocentric thrusting periods, the thrust is directed optimally as determined by a calculus-of-variations computer program. (Hereinafter, this thrust program is referred to as constant thrust.) During spirals, the thrust is directed tangentially (ref. 7), which very closely approximates the true optimum (ref. 1). The entire mission is treated as a series of two-body problems, and, for the heliocentric transfer, only the optimum travel angle transfer is used. In all cases, the heliocentric transfer is made between assumed circular, coplanar orbits at a mean distance from the Sun. For circular planet orbits, the optimum travel angle and best encounter are synonymous.

In this report, the propellant fraction is given as a function of effective jet power to initial weight ratio for constant values of specific impulse and total travel time for Mars and Venus orbiting probes. From this data, the effects of any vehicle parameters on payload can be investigated. Since thrustor efficiency varies widely with design and type, no attempt other than an illustrative example has been made to generalize the effect of their efficiencies on the payload. The propellant fraction data given form a sufficiently complete starting point for most mission analyses, vehicle design, and thrustor evaluation.

To illustrate the use of the data, the performance of a typical Mars orbiting probe is discussed in appendix B. The problem treated is that of finding maximum payload for a typical spacecraft. The effect of electric power and gross weight are discussed to illustrate problems encountered in integrating the spacecraft with an orbital booster.



Assumed planet orbits	Radius, m	Polar angular velocity[a], radians/sec
Venus	1.0814×10^{11}	3.2364×10^{-7}
Earth	1.4950	1.9910
Mars	2.2779	1.0586

[a]Based on solar gravitational constant $\mu = 1.3245 \times 10^{20}$ m^3/sec^2.

Figure 1. - Schematic of Mars and Venus orbiting probes.

ANALYSIS

Orbiting Probe Trajectory

In any mission analysis work, some criterion is chosen for optimization. This generally is the maximum useful payload fraction commensurate with such factors as economy, reliability, availability, and so forth. Even when the latter factors are neglected, it is desirable to employ minimum propellant maneuvers from the start to the end of the mission. In some cases, however, optimum trajectories and their corresponding thrust programs are not compatible with available thrustors and guidance systems.

For Mars and Venus orbiting probes using early electric propulsion systems, an optimum on-off constant-thrust trajectory appears feasible; that is, optimum in the sense that thrust vector steering over the relatively long propulsion periods gives minimum propellant expenditure. Such interplanetary trajectories with optimum placement and duration of intermediate coast phases have been demonstrated in references 3 and 4. These trajectories, computed by indirect variational calculus techniques, have been used herein for the interplanetary phase of the mission. The entire trajectory (fig. 1) for the orbiting probe is treated as a series of two-body problems - an Earth escape spiral, a heliocentric transfer, and a planetocentric capture spiral. The planets (Venus, Earth, and Mars) are assumed to be in circular, coplanar orbits about the Sun. The same thrust and effective specific impulse is assumed operative over the entire trajectory. The spirals are constant tangential thrust maneuvers between 400-statute-mile circular orbits and escape relative to the planets (ref. 7). Although the restriction of a 400-statute-mile orbit is arbitrary, the effect of initial orbit altitude on the overall mission is small. This effect is further discussed in the section RESULTS AND DISCUSSION for a typical set of vehicle parameters.

In basic trajectory work, two parameters are important - the thrust acting on the vehicle F and the rate of change of vehicle mass $\dot{m}_t$; however, results are not always conveniently expressed nor widely used with F and $\dot{m}_t$ as parameters. Other trajectory performance parameters more widely used in electric propulsion mission studies are effective specific impulse

$$I_{eff} \equiv \frac{F}{g_c \dot{m}_t} \tag{1}$$

and effective jet power

$$P_{jeff} \equiv \frac{g_c I_{eff} F}{2} \tag{2}$$

where $g_c = 9.80665$ meters per second per second.

The terminology of effective specific impulse and effective jet power is used here to emphasize the fact that $\dot{m}_t$ is the total mass flow rate. In an ion thrustor system, for example, $\dot{m}_t$ can represent the accelerated ions that produce the thrust and neutral atoms, which result from the inefficiency of ionization. Note that the assumption of constant F and I_{eff} also implies constant $\dot{m}_t$.

In addition to the two trajectory performance parameters given previously, the mission parameter total travel time T_t is also important in that it affects the propellant fraction and mission reliability. Thus, propellant fraction can be stated as

$$w_p = w_p\left(\frac{P_{jeff}}{W_0}, I_{eff}, T_t\right) \tag{3}$$

for the optimum constant-thrust trajectories.

Weight Analysis

To this point only the trajectory has been discussed. The purpose of this section is to illustrate how the w_p data is used in the orbiting probe mission analysis. The important parameters affecting payload fraction are defined, and criteria are given for maximum w_L. Since this is a preliminary analysis, no attempt is made to define a useful payload by including the multitude of small systems that make up a spacecraft. Hereinafter the spacecraft is considered to be payload, electric powerplant, propellant, and a propellant-dependent structure.

Definition of parameters. - With the aforementioned assumptions,

$$W_0 = W_L + (1 + k_s)W_p + W_{pp} \tag{4}$$

or

$$w_L = 1 - (1 + k_s)w_p - w_{pp} \tag{5}$$

where the W's are system weights and w's are weight fractions. For the optimized constant-thrust trajectories, the propellant fraction is given by equation (3). The weight fraction of the powerplant is defined as

$$w_{pp} \equiv \frac{\alpha \mathcal{P}}{W_0} \tag{6}$$

where α is the specific weight of an electric powerplant delivering $\mathcal{P}$ kilowatts of electric power to the thrust producing system. The weight of the powerplant is assumed to be the weight of all the components (heat sources, conversion equipment, conditioning equipment, etc.) necessary to produce electric power at the required currents and voltages.

If the overall efficiency η is defined as the ratio of effective jet power to input power, the powerplant fraction becomes

$$w_{pp} = \frac{\alpha}{\eta} \frac{P_{jeff}}{W_0} \tag{7}$$

In appendix B, it is shown that for ion thrustors the overall efficiency can be expressed as the product of the propellant utilization efficiency η_u and the thrustor power efficiency η_P. It is also shown that η_P is only a function of η_u and I_{eff} for a given propellant and thrustor design. Therefore,

$$w_{pp} = \frac{\alpha}{\eta(\eta_u, I_{eff})} \frac{P_{jeff}}{W_0} \tag{8}$$

Substituting equations (3) and (8) into equation (5) results in the payload fraction

$$w_L = 1 - (1 + k_s)w_p\left(\frac{P_{jeff}}{W_0}, I_{eff}, T_t\right) - \frac{\alpha}{\eta(\eta_u, I_{eff})}\frac{P_{jeff}}{W_0} \tag{9}$$

Criteria for maximum w_L. - From equation (9), it is evident that the parameters affecting the payload fraction are the vehicle performance parameters P_{jeff}/W_0 and I_{eff}, the thrustor performance parameters η_u, the mission parameters T_t, and the constants k_s and α. These parameters can be divided into two groups - those free for optimization and those that must be treated as specified constants. If T_t, k_s, and α are treated as specified constants, then payload fraction can be optimized with respect to P_{jeff}/W_0, I_{eff}, and η_u. A range of the constants will then give their gross effects on an optimized payload fraction. Thus, differentiating equation (9) for this case gives

$$dw_L = \left[\frac{\alpha\left(\frac{P_{jeff}}{W_0}\right)}{\eta^2}\frac{\partial\eta}{\partial\eta_u}\right]d\eta_u - \left[(1 + k_s)\frac{\partial w_p}{\partial\left(\frac{P_{jeff}}{W_0}\right)} + \frac{\alpha}{\eta}\right]d\left(\frac{P_{jeff}}{W_0}\right)$$

$$- \left[(1 + k_s)\frac{\partial w_p}{\partial I_{eff}} - \frac{\alpha\left(\frac{P_{jeff}}{W_0}\right)}{\eta^2}\frac{\partial\eta}{\partial I_{eff}}\right]dI_{eff} \tag{10}$$

For a maximum w_L, $dw_L = 0$, and, since P_{jeff}/W_0, I_{eff}, and η_u are all independent variables, coefficients of the differentials of these variables must independently be zero. If any one of these independent variables is also considered specified, then its differential is zero and no information is obtained from the coefficient in equation (10). In general, the necessary conditions for a maximum payload fraction are

$$\eta_u: \qquad \frac{\partial\eta}{\partial\eta_u} = 0 \tag{11a}$$

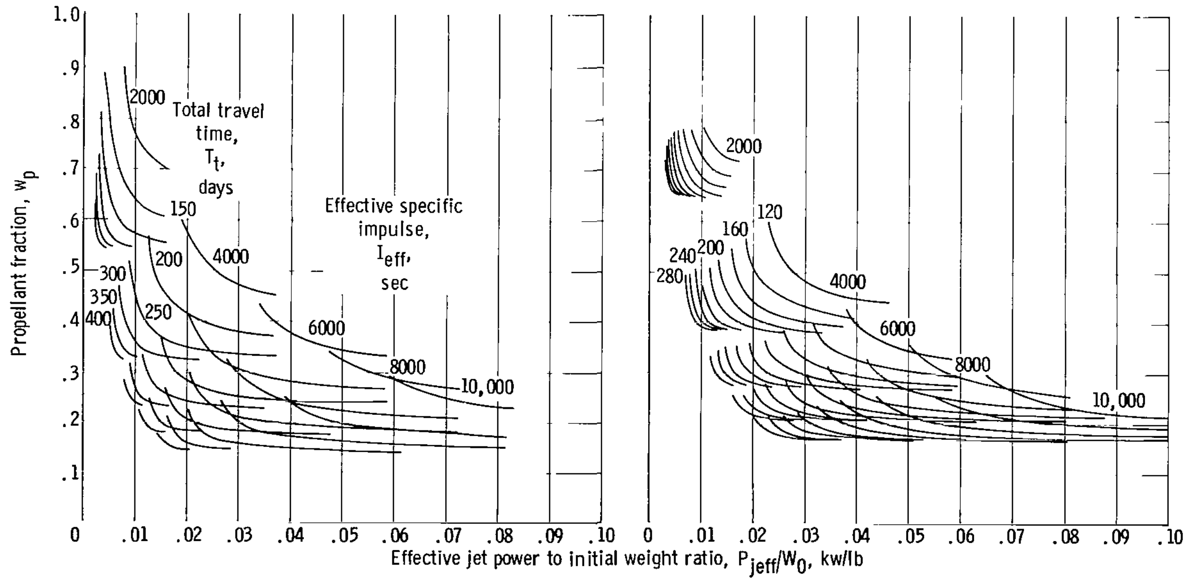
$$I_{eff}: \qquad \frac{\partial w_p}{\partial I_{eff}} = \frac{\alpha\left(\frac{P_{jeff}}{W_0}\right)}{(1 + k_s)\eta^2}\frac{\partial\eta}{\partial I_{eff}} \tag{11b}$$

$$\frac{P_{jeff}}{W_0}: \qquad \frac{\partial w_p}{\partial\left(\frac{P_{jeff}}{W_0}\right)} = -\frac{\alpha}{(1 + k_s)\eta} \tag{11c}$$

Since the condition expressed by equation (11a) is simply the requirement of maximum overall efficiency for each I_{eff}, using $\eta = \eta_{max}(I_{eff})$ satisfies one condition for maximum w_L. The details of satisfying equation (11a) are given in appendix B for the Mars probe using state-of-the-art mercury electron-bombardment thrustors. This conclusion could have been made directly from equation (9) since maximum overall efficiency ensures minimum w_p and has no further effect on w_p at a given I_{eff}. The propellant utilization efficiency is, however, used as an independent variable because such component weights as thrustors (if they had been included) may require overall efficiencies less than maximum to ensure maximum w_L.

At each P_{jeff}/W_0, a local maximum can be obtained with respect to I_{eff}. The condition necessary for the maximum is expressed by equation (11b), which depends on the efficiency function, the specific powerplant weight, and the structural factor. Thus, the choice of the best I_{eff} is affected by all these parameters; however, for the special case when efficiency is assumed a constant (not a function of I_{eff}), $\partial\eta/\partial I_{eff} = 0$. Hence $\partial w_p/\partial I_{eff} = 0$ at maximum payload fraction. For this case, the optimum I_{eff} is independent of any of the specified constants. Moreover, the propellant fraction is a minimum with respect to I_{eff}.

At a given I_{eff}, equation (11c) is the condition necessary for a local maximum with respect to P_{jeff}/W_0. Since this expression is independent of P_{jeff}/W_0, it is generally easy to satisfy. For $\eta = 1.0$ and $k_s = 0$, the condition reduces to the requirement that the slope equals $-\alpha$.



(a) Mars orbiting probe.

(b) Venus orbiting probe.

Figure 2. - Effect of vehicle performance parameters and total travel time on propellant fraction.

To summarize, two methods can be used to obtain maximum w_L. It can be obtained directly by computing several values, plotting the results, and selecting the maximum. Another method is the indirect method of satisfying the criteria for a maximum (i.e., eq. (11)) to determine the optimum performance parameters. These can then be used to compute the maximum w_L. Both methods have merit; the former is straightforward but can involve many repeated calculations, while the latter method can lead to reduced computations and often gives an insight about the nature of the optimum. The utility of this method is illustrated in appendix B.

RESULTS AND DISCUSSION

Trajectory Results

The results of the orbiting probe trajectory calculations are given in figure 2. In this figure, w_p is given as a function of P_{jeff}/W_0 (kw/lb) with lines of constant I_{eff} (sec) and T_t (days). A typical curve is given in figure 3 to aid in the explanation of figure 2. Note in figure 3 that at a given I_{eff} and T_t, w_p rapidly decreases from the all propulsion (no coast) value to the value characterized by the best heliocentric travel time. This lower bound is the low-acceleration equivalent of the Hohmann transfer for impulsive thrust. Not all of the curves of figure 2 end at this lower bound because an arbitrary range on F/W_0 was imposed - values less than 0.5×10^{-4} or greater than 5×10^{-4} were not considered.

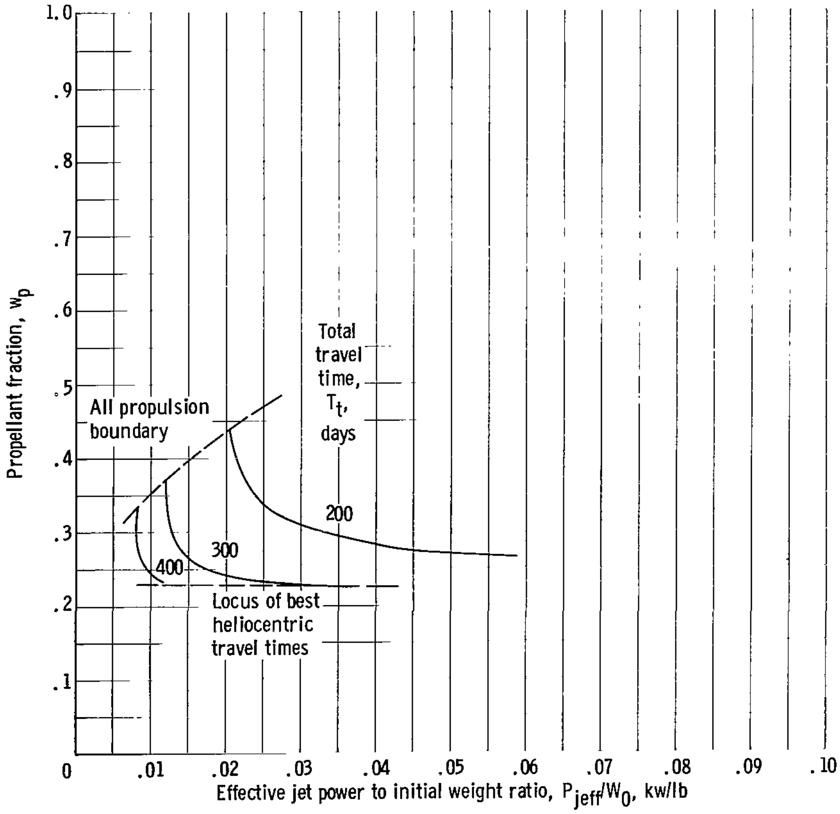


Figure 3. - Typical propellant fraction curves for Mars orbiting probe. Effective specific impulse, 6000 seconds.

Several characteristics are to be noted about the curves. First, the clusters of T_t curves at one I_{eff} all approach nearly the same value of w_p at the low-acceleration equivalent of the Hohmann transfer. For example, at I_{eff} = 2000 seconds, the lowest w_p for all T_t is about 0.54 for the Mars orbiting probes. Second, P_{jeff}/W_0, slightly greater than the all-propulsion-value, results in a large reduction in w_p. This effect is equivalent to allowing for a coasting phase on the heliocentric trajectory.

A comparison of figures 2(a) and (b) reveals that propellant fraction at low values of P_{jeff}/W_0 and I_{eff} are significantly higher for Venus missions. This penalty is due to the higher gravity losses experienced during the Venus capture spiral since the mass of Venus is approximately 7.5 times the mass of Mars. At higher values of P_{jeff}/W_0 and I_{eff}, this difference vanishes and the Mars mission is seen to be more difficult. The predominant effect is that the Mars-Earth radius ratio (~1.52, i.e., relative distances from the Sun) is higher than the Earth-Venus radius ratio (~1.38). If, however, the Mars probe is given slightly more time, the missions are much more comparable. For example, an orbiting probe vehicle with $P_{jeff}/W_0 = 0.04$ kilowatt per pound, $I_{eff} = 6000$ seconds, and $w_p = 0.328$ can deliver the same mass to Mars in 175 days or to Venus in 140 days.

Effect of Initial Orbit Altitude

As stated previously, the effect of initial orbit altitude on the mission is small. This is illustrated in figure 4 where w_p is plotted against initial orbit altitude h_0 for a 250-day Mars mission with $I_{eff} = 6000$ seconds. From the figure at $P_{jeff}/W_0 = 0.020$ kilowatt per pound, it is seen that for a change in h_0 from 200 to 1000 miles, w_p decreases from 0.283 to 0.270 - a savings of only 4.6 percent. The effect of h_0 is small because a vehicle at escape (the start of the heliocentric transfer) from a low orbit has a higher thrust acceleration than the same vehicle from a higher orbit. This higher thrust acceleration tends to reduce propellant requirements for the remainder of the trip; however, a spiral from a low orbit requires more time. With the constraint of constant total travel time, this means less time for the heliocentric transfer and capture spiral and for this case, more propellant. As the Mars spiral requirements are small, these effects primarily influence the heliocentric transfer. Thus by trading allotted time and thrust to weight, the heliocentric propellant requirements remain about the same, and the overall effect is approximately the propellant difference for the Earth spiral. From these arguments it can be concluded that there is little error in using the data for missions that commence in circular orbits somewhat different than the assumed value of 400 statute miles. It is also believed that initial orbit altitude has little effect on the Venus orbiting probes.

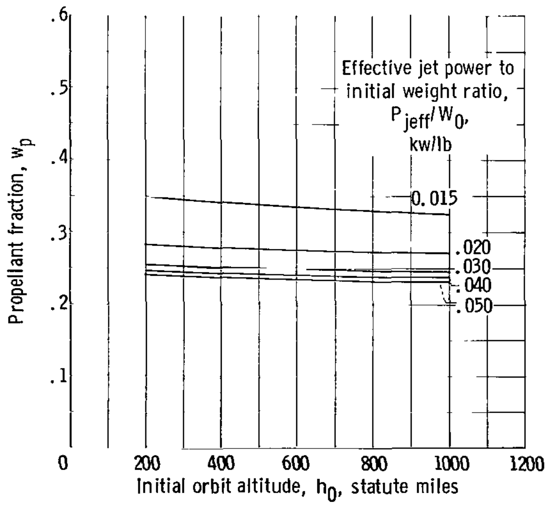


Figure 4. - Effect of initial orbit altitude on propellant fraction for Mars orbiting probes. Total travel time, 250 days; effective specific impulse, 6000 seconds.

Effect of Performance Parameters

In the ANALYSIS, it was shown that for maximum w_L minimum propellant fraction was optimum for constant efficiency. In the example in appendix B, equation (11b) was satisfied for an efficiency function, which is representative of mercury electron-bombardment thrustors. The results were also shown to be

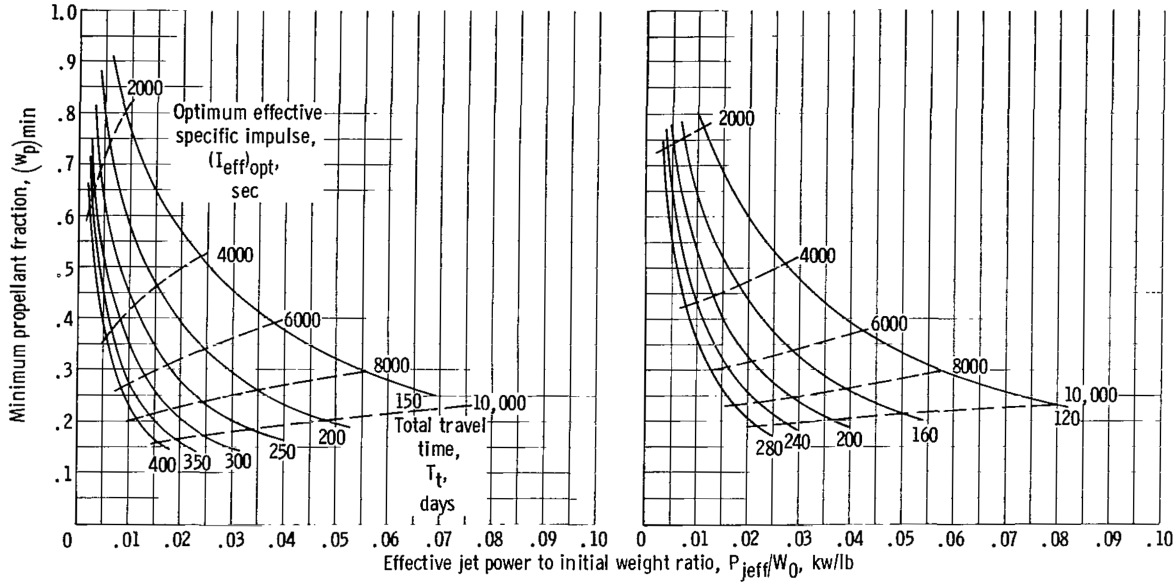


Figure 5. - Minimum propellant fraction as function of effective jet power to initial weight ratio, total travel time, and optimum effective specific impulse.

reasonably approximated by assuming $\partial w_p/\partial I_{eff} = 0$. Therefore, in the following payload computations, η is assumed to be constant. In figure 5, minimum

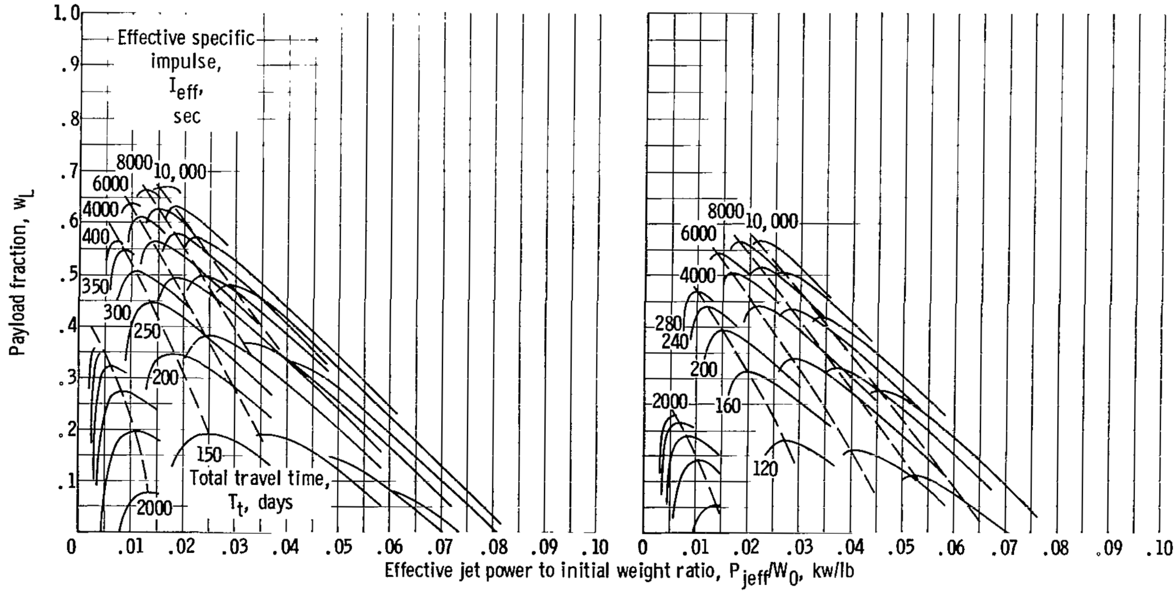
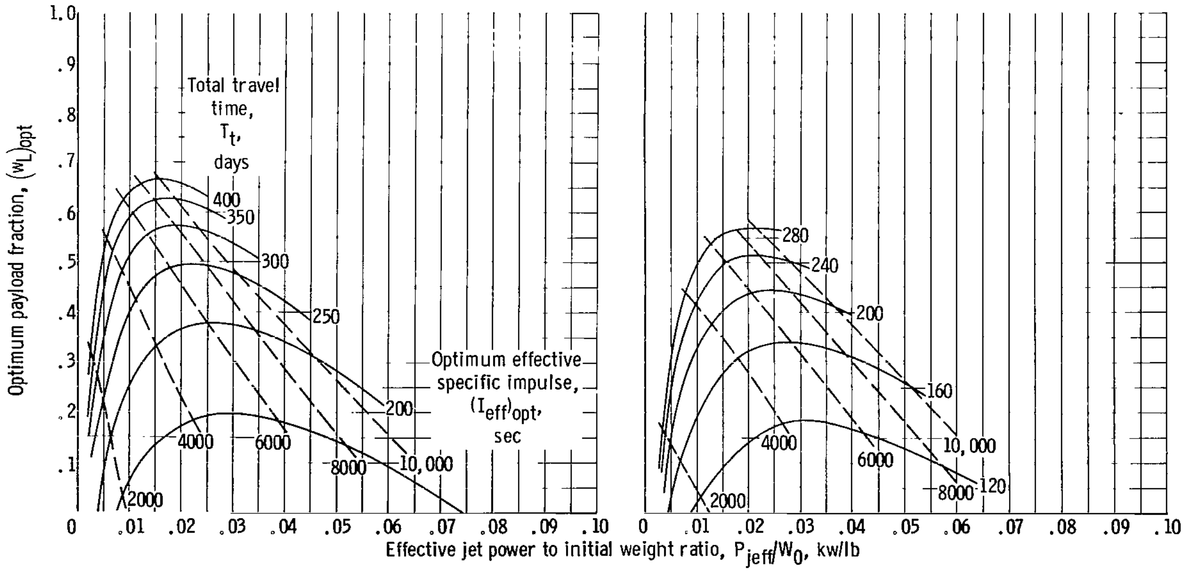


Figure 6. - Effect of vehicle performance parameters and total travel time on payload fraction. Specific powerplant weight, 10 pounds per kilowatt; structural factor, 0.10.

propellant fraction is given as a function of P_{jeff}/W_0 for values of constant T_t. The optimum values of I_{eff} are also traced on the figure. With the assumption of constant efficiency, it is convenient to define $\alpha' \equiv \alpha/\eta$ since α and η appear only as the ratio in equations (9) and (11).

To illustrate the optimum discussed and the effects of off-optimum vehicle performance parameters, w_L is given in figure 6 for $\alpha' = 10$ pounds per kilowatt and $k_s = 0.10$. The dashed lines tie the points where w_L is optimized with respect to P_{jeff}/W_0 at constant I_{eff}. At any value of T_t and I_{eff} (e.g., 250 days and 6000 sec for the Mars mission), w_L is seen to be relatively insensitive to P_{jeff}/W_0 near the optimum of 0.0182 kilowatt per pound. For example, a 10-percent change in P_{jeff}/W_0 produces about the same change in w_L; however, a random choice of P_{jeff}/W_0 can result in very significant penalties. If P_{jeff}/W_0 is held constant, the effects of I_{eff} are roughly the same as those of P_{jeff}/W_0. Thus, optimizing both parameters is equally important.

If the envelope curve is drawn to the curves of constant T_t for the range of I_{eff}, maximum w_L would be obtained as a function of P_{jeff}/W_0. These are equivalently the payload fractions for the cases of minimum w_p. The results for the complete range of T_t are given in figure 7. As shown in figure 7, I_{eff} can be varied from the optimum in the range of 5000 to 10,000 seconds with little penalty in w_L. Also note that the maximum w_L attainable is determined mainly by T_t. An interesting result, also noted in reference 6, is that P_{jeff}/W_0 and the optimum I_{eff} at any T_t vary so that the initial thrust to weight ratio F/W_0 is nearly a constant. For example, at



(a) Mars orbiting probes.

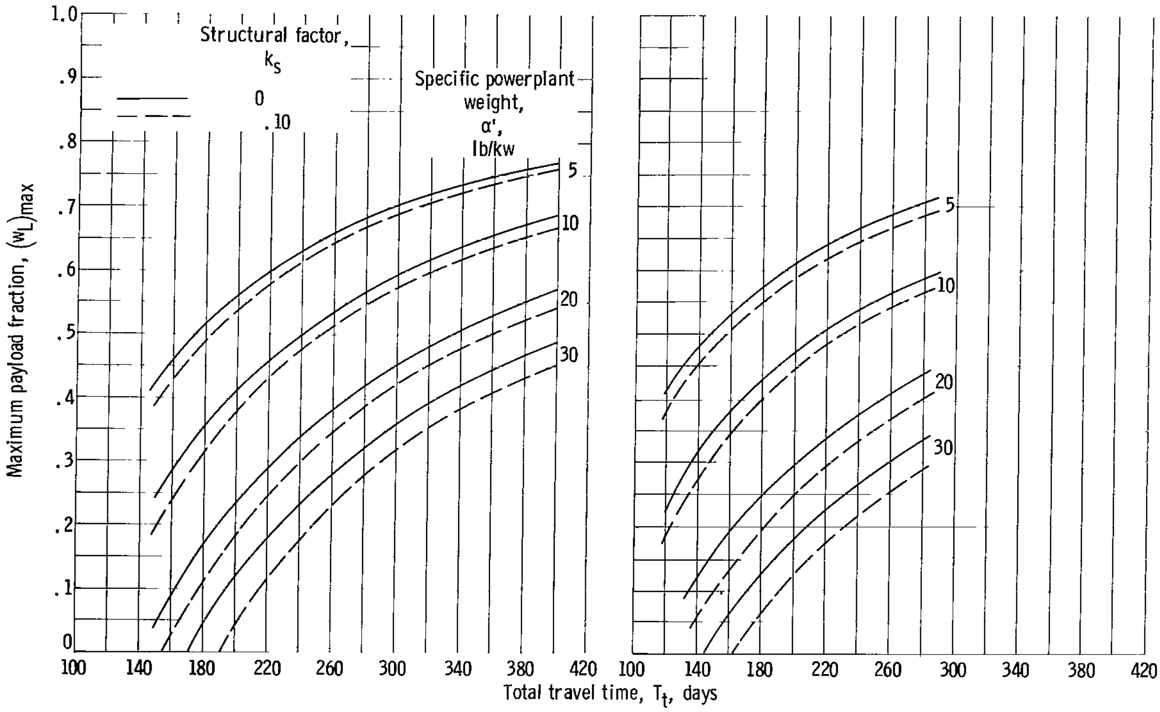
(b) Venus orbiting probes.

Figure 7. - Optimum payload fraction as function of effective jet power to initial weight ratio, total travel time, and optimum effective specific impulse. Specific powerplant weight, 10 pounds per kilowatt; structural factor, 0.10.

T_t = 250 days, $F/W_0 = 1.28\times10^{-4}$ at I_{eff} = 4000 seconds and 1.42×10^{-4} at I_{eff} = 10,000 seconds.

Effect of Specific Powerplant Weight, Structural Factor, and Total Travel Time

Maximum w_L. - In figure 8 the maximum payload fraction is plotted as a function of total travel time. For the chosen values of α' and k_s, both



(a) Mars orbiting probes.

(b) Venus orbiting probes.

Figure 8. - Effect of total travel time, specific powerplant weight, and structural factor on maximum payload fraction.

P_{jeff}/W_0 and I_{eff} have been optimized. It is noted that both α' and T_t have a much larger effect than k_s. Also, a Mars orbiting probe with a powerplant at α' = 30 pounds per kilowatt can carry about the same payload as one with α' = 10 pounds per kilowatt if T_t is extended by 100 days.

Optimum P_{jeff}/W_0. - In figure 9, P_{jeff}/W_0, to achieve maximum payload fraction, is plotted against T_t. This figure shows that the optimum P_{jeff}/W_0 is primarily a function of α'. This is particularly true at high values of α' where only a small change in P_{jeff}/W_0 occurs over the entire range of T_t. It is also noted that at low α', the P_{jeff}/W_0 is high and vice versa at high α'. In fact, the powerplant fraction $w_{pp} = \alpha' P_{jeff}/W_0$ is roughly a constant between 1/4 and 1/3 over the entire range of T_t and α', except at high T_t and low α' and low T_t and high α' (i.e., the upper left-hand corner and lower right-hand corner of fig. 9).

From figure 9, it is also seen that decreasing the total travel time requires increasing P_{jeff}/W_0. For the low values of α', the increase is more

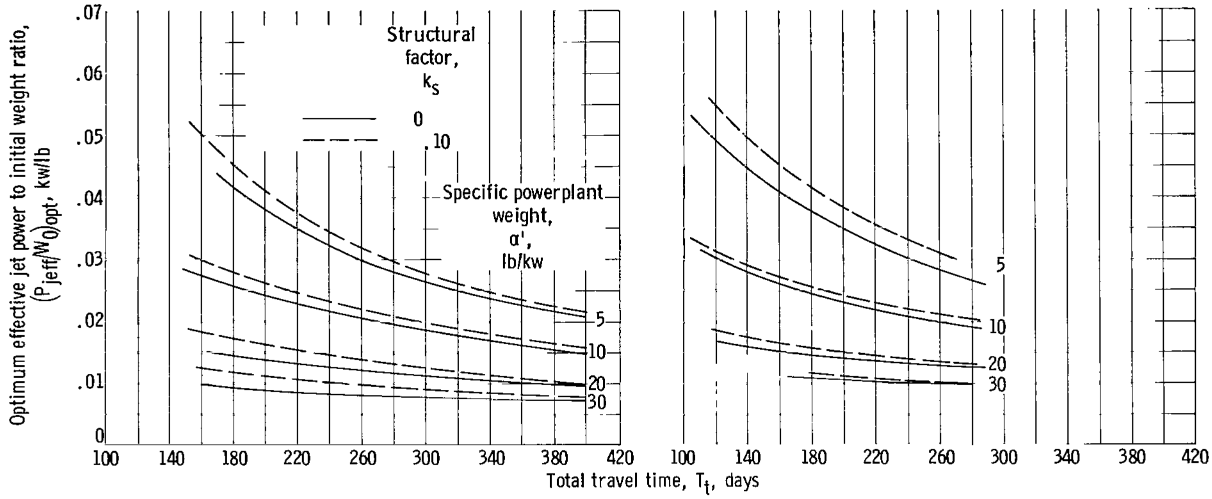


Figure 9. - Effect of total travel time, specific powerplant weight, and structural factor on optimum effective jet power to initial weight ratio.

pronounced since propellant fraction can be reduced without significant increases in the powerplant weight. At any α' the 10-percent structural factor has the constant effect of increasing P_{jeff}/W_0 by about 0.002 kilowatt per pound.

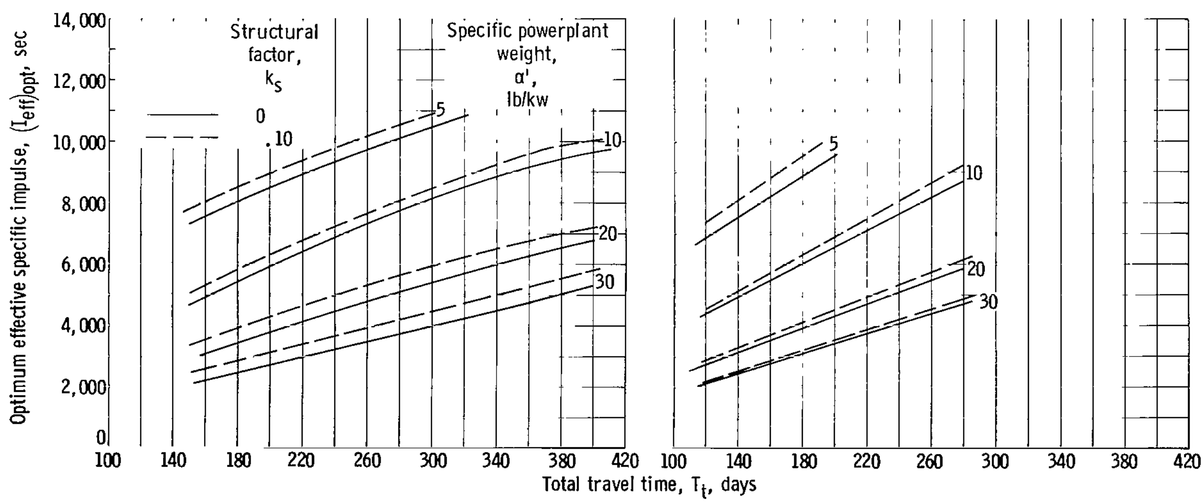


Figure 10. - Effect of total travel time, specific powerplant weight, and structural factor on optimum effective specific impulse.

Optimum I_{eff}. - Figure 10 gives the optimum I_{eff} for the maximum payload cases. The effect of T_t here is opposite to that of P_{jeff}/W_0 because the vehicle acceleration is directly proportional to P_{jeff}/W_0 and inversely proportional to I_{eff}. Specific powerplant weight α' also determines the range of I_{eff}. For example, the optimum I_{eff} is between 5000 and 10,000 seconds for $\alpha' = 10$ pounds per kilowatt and between 2000 and 5000 seconds for $\alpha' = 30$ pounds per kilowatt. The effect of the 10-percent structural factor is to increase the optimum I_{eff} by about 500 seconds or less. From figures 9 and 10, the optimum F/W_0 can be obtained. If this is done, it will be seen that F/W_0 is primarily determined by T_t. The effect of increasing T_t is to decrease the F/W_0. Increasing α' has the slight effect of reducing the optimum F/W_0.

CONCLUDING REMARKS

A parametric study has been made of constant-thrust, low-acceleration Mars and Venus orbiting probes. Propellant fractions are given for a broad range of vehicle performance parameters for the missions that are treated as a series of two-body problems. Constant tangential thrust is used for the planetocentric portion, and an optimum constant thrust (with coasting periods) is used for the heliocentric portion of the mission. Although all the data is for a mission commencing in a 400-statute-mile circular orbit, the effect of initial orbit altitude is shown to be small.

The propellant fraction data for both Mars and Venus is very comparable for the same set of vehicle performance parameters (P_{jeff}/W_0 and I_{eff}) except that the Venus mission occurs in less time. This is true because the stringent requirements of the Venus capture spiral tend to compensate for the difference between Mars-Earth and Earth-Venus radius ratios.

Payload fractions are given for a simplified model of an electrically propelled spacecraft. The effect of off-optimum vehicle performance parameters is illustrated for a representative set of parameters over a wide range of travel time. The effects of specific powerplant weight, a structural factor, and total travel time are illustrated for maximum payload fraction where P_{jeff}/W_0 and I_{eff} are optimized.

As these results were all for the case of constant thrustor efficiency, a representative variation of η with I_{eff} was studied. The results (in appendix B) indicate that the optimum propellant fraction is only slightly higher than the minimum propellant fraction, and that the overall effect is essentially only an increased powerplant fraction. The corresponding vehicle performance parameters are slightly lower for this case than for the case of maximum payload fraction with 100-percent thrustor efficiency.

In appendix B the problem of maximum W_L is discussed. It is shown that when W_0 is specified, the case of maximum W_L with respect to P is identical to the case of maximum payload fraction. In general, maximum payload at any W_0 and P are not maximum payload fraction cases. These considerations are important when a spacecraft of given power is integrated with available boosters of a given orbital payload capabilities.

Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio, August 4, 1964

APPENDIX A

SYMBOLS

a	thrust acceleration
F	thrust, newtons
F/W_0	initial acceleration, Earth g's
g_c	9.80665 m/sec^2
h	orbit altitude, statute miles
I_{eff}	effective specific impulse, sec
k	a units conversion constant
k_s	propellant dependent structural factor
M	molecular weight
$\dot{m}$	mass flow rate of accelerated propellant, kg/sec
$\dot{m}_t$	total mass flow rate of vehicle, kg/sec
$\mathcal{P}$	input power to thrustors, w
P_j	jet power, w
P_{jeff}	effective jet power, w
P_l	thrustor power losses, w
T	travel time, days
$\bar{v}$	average exhaust velocity, m/sec
W	system weight, lb
w	system weight fraction
α	powerplant specific weight, lb/kw
α'	α/η, lb/kw
η	overall thrustor efficiency
η_p	thrustor power efficiency

η_u propellant utilization efficiency

ξ thrustor power loss per ion produced, ev/ion

Subscripts:

L payload

max maximum

0 initial

opt optimum

p propellant

pp powerplant

t total

APPENDIX B

PAYLOAD CAPABILITIES OF TYPICAL

MARS ORBITING PROBE

The purpose herein is to discuss the performance of a low-thrust Mars orbiting probe and to illustrate the effect of a typical state-of-the-art engine efficiency. Since subsystems such as thrustor and powerplant are often developed independently, the principal problem is one of integrating and assessing their individual effects on the mission. Suppose the problem is stipulated as that of optimally delivering payload to Mars in 300 days with a nuclear-electric powerplant weighing 10 pounds per kilowatt. Mercury electron-bombardment ion thrustors are to be used, and the effects of their efficiency on performance are to be estimated. This problem is similar to the maximum payload fraction problem in the text except for the efficiency function assumed, and the performance can readily be assessed from the data and methods given. If, however, the electric powerplant output is fixed and the system is to be integrated with a booster with given payload capability, maximum payload does not necessarily occur at the maximum payload fraction. In other words, the problem would then be to maximize the delivered payload of an electrically propelled vehicle, given an electric powerplant output and some gross weight in orbit determined by the booster performance. This case, termed the problem of maximum payload, will also be treated herein.

Effects of Thrustor Efficiency

To assess the effects of thrustor efficiency, it is necessary to determine how the overall efficiency is affected by vehicle performance parameters and thrustor performance parameters. The overall efficiency of the thrustor is defined as the ratio of effective jet power to total input power

$$\eta \equiv \frac{P_{j\,eff}}{\mathscr{P}} \tag{B1}$$

When any thrustor is considered (ref. 8), the power efficiency can be defined as the ratio of jet power to total input power

$$\eta_P \equiv \frac{P_j}{\mathscr{P}} \tag{B2}$$

where P_j is defined by the net thrust and average exhaust velocity as follows:

$$P_j \equiv \frac{1}{2} F\overline{v} \tag{B3}$$

The average exhaust velocity $\overline{v}$ is defined as the thrust divided by the flow rate of the accelerated propellant $\dot{m}$. If the propellant utilization efficiency

η_u is defined as the ratio of accelerated propellant to the total mass flow rate $\dot{m}_t$, the average exhaust velocity becomes

$$\bar{v} \equiv \frac{F}{\eta_u \dot{m}_t} \tag{B4}$$

where

$$\eta_u \equiv \frac{\dot{m}}{\dot{m}_t} \tag{B5}$$

When equations (B3) and (B4) are substituted into equation (B2), the power efficiency becomes

$$\eta_P = \frac{1}{2} \frac{F^2}{\eta_u \dot{m}_t \mathscr{P}} \tag{B6}$$

If equations (1) and (2) of the ANALYSIS are combined,

$$P_{jeff} = \frac{1}{2} \frac{F^2}{\dot{m}_t} \tag{B7}$$

Thus, equations (B6) and (B7) give

$$\eta_P = \frac{P_{jeff}}{\eta_u \mathscr{P}} \tag{B8}$$

and comparing equations (B8) and (B1) shows that

$$\eta = \eta_P \eta_u \tag{B9}$$

Thus, it is seen that the overall efficiency is the product of the power and propellant utilization efficiencies. Another expression for the power efficiency can be developed if total input power is written as

$$\mathscr{P} = P_j + \sum P_l \tag{B10}$$

where $\sum P_l$ is the sum of all power losses. The sum of power losses can be expressed as a power loss per ion produced ξ multiplied by the ion flow rate. For a plane diode, electron-bombardment thrustor with all molecules singly charged, the sum of the power losses is (ref. 8)

$$\sum P_l = \frac{k \xi \eta_u \dot{m}_t}{M} \tag{B11}$$

where k is a constant for the conversion of units and M is the molecular weight of the propellant. The power efficiency is then

$$\eta_P = \frac{P_j}{\mathscr{P}} = \frac{1}{1 + \frac{2k\xi}{M\bar{v}^2}} = \frac{1}{1 + \frac{2k\xi\eta_u^2}{g_c^2 M I_{eff}^2}} \tag{B12}$$

Thus, the overall efficiency is

$$\eta = \eta_u \eta_P = \frac{\eta_u}{1 + \frac{2k\xi\eta_u^2}{g_c^2 M I_{eff}^2}} \tag{B13}$$

From equation (B13) it is evident that the overall efficiency is a function of the propellant type, the propellant utilization efficiency, the power loss per ion, and the effective specific impulse. Furthermore, if a particular engine design and propellant are considered, an operating curve of the power loss per ion as a function of propellant utilization efficiency can be experimentally determined, and η can be expressed as

$$\eta = \eta(\eta_u, I_{eff}) \tag{B14}$$

With the relations for η developed and the criteria for maximum payload fraction developed in the ANALYSIS, the effect of efficiency can be determined. A first condition $\partial\eta/\partial\eta_u = 0$ (eq. (11a)) is satisfied by differentiating equation (B13). Thus, for $\partial\eta/\partial\eta_u = 0$,

$$\frac{d\xi}{d\eta_u} = \frac{\frac{Mg_c^2}{2k} I_{eff}^2 - \xi\eta_u^2}{\eta_u^3} \tag{B15}$$

which at a given I_{eff} defines a point on the operating curve (ξ against η_u). This point corresponds to the maximum overall efficiency at the given I_{eff}. The operating curve assumed here is given in figure 11 (ref. 9). For this case, maximum overall efficiency, as defined by equation (B15), is given in figure 12 as a function of I_{eff}. Thus, using this

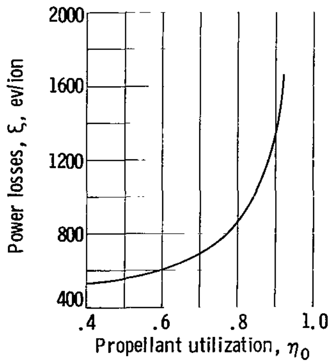


Figure 11. - Power losses in mercury electron-bombardment thrustors.

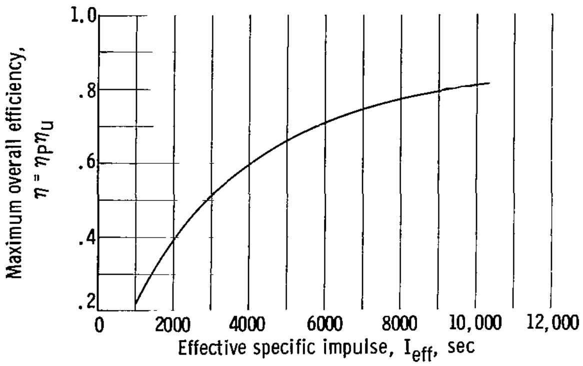


Figure 12. - Maximum overall efficiency of mercury electron-bombardment thrustors.

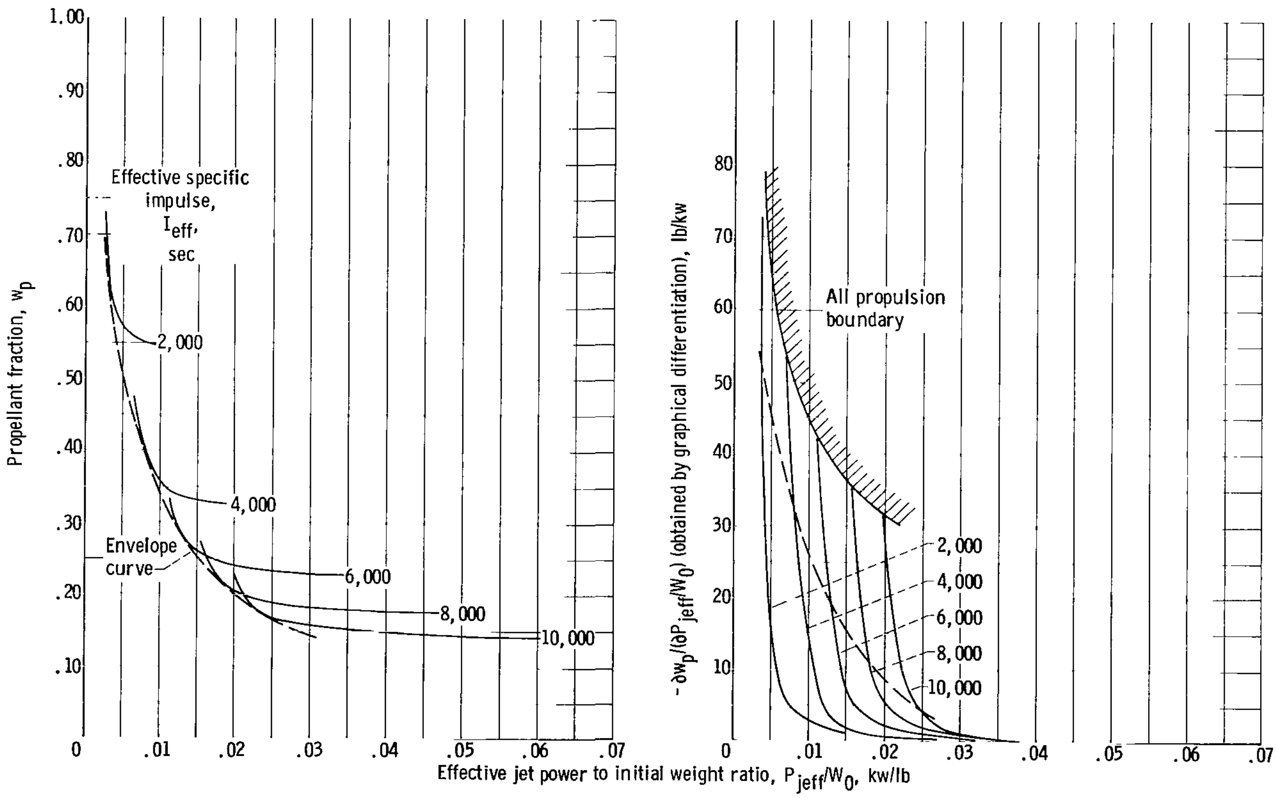
figure to calculate w_L satisfies one condition for a maximum.

Two conditions remain to be satisfied to obtain an overall maximum w_L. They are the conditions related to P_{jeff}/W_0 and I_{eff}. From the ANALYSIS, equations (11b) and (11c) become

$$\frac{P_{jeff}}{W_0}: \qquad \frac{\partial w_p}{\partial\left(\frac{P_{jeff}}{W_0}\right)} = -\frac{\alpha}{(1 + k_s)\eta} \tag{B16a}$$

$$I_{eff}: \qquad \frac{\partial w_p}{\partial I_{eff}} = \frac{\alpha\left(\frac{P_{jeff}}{W_0}\right)}{(1 + k_s)\eta^2} \frac{\partial \eta}{\partial I_{eff}} \tag{B16b}$$

To obtain the overall maximum w_L, both conditions must be simultaneously satisfied. This can be achieved by a systematic trial-and-error procedure of



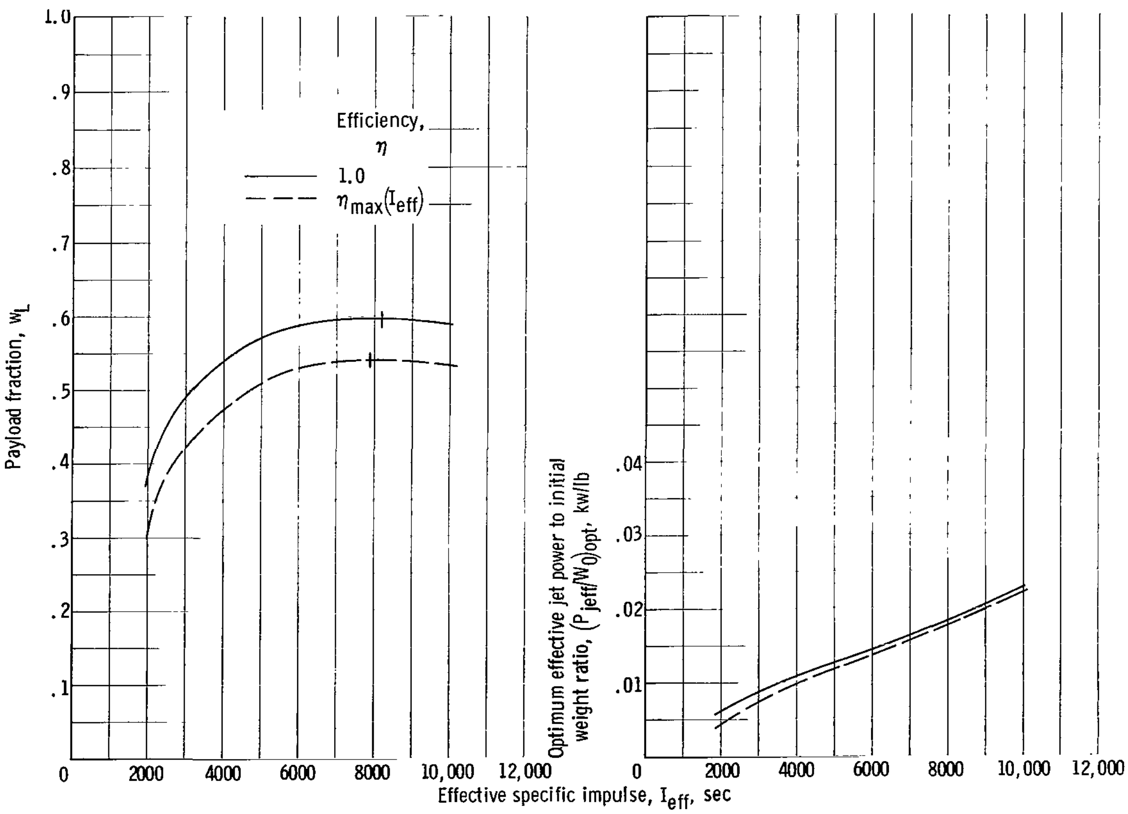
(a) Propellant fraction as function of vehicle performance parameters.

(b) Partial derivative of propellant fraction with respect to effective jet power to initial weight ratio as function of vehicle performance parameters.

Figure 13. - Trajectory results for 300-day Mars orbiting probe.

finding the optimum P_{jeff}/W_0 and I_{eff}, or by satisfying either one of equations (B16) over the range of the other parameter and plotting the results to determine the overall maximum. The latter procedure, with equation (B16a) satisfied over the range of I_{eff} was used here to illustrate the effect of thrustor efficiency. In this way local maximum with respect to P_{jeff}/W_0 are compared over the entire range of I_{eff}.

The example chosen is a 300-day Mars orbiting probe with $\alpha = 10$ pounds per kilowatt and $k_s = 0$. The propellant fraction data and the auxiliary plot of $\partial w_p/\partial(P_{jeff}/W_0)$ are given in figure 13. From this data, the results obtained for the efficiency comparison are shown in figure 14. Figure 14(a) gives w_L as a function of I_{eff}, and figure 14(b) gives the corresponding optimum values of P_{jeff}/W_0 over the range of I_{eff}. Note that at maximum w_L (fig. 14(a)) the decrease in efficiency causes a decrease in I_{eff} and P_{jeff}/W_0. In this case the optimum I_{eff} decreases from 8150 to 7900 seconds and the optimum P_{jeff}/W_0 decreases from 0.0185 to 0.0175 kilowatt per pound.

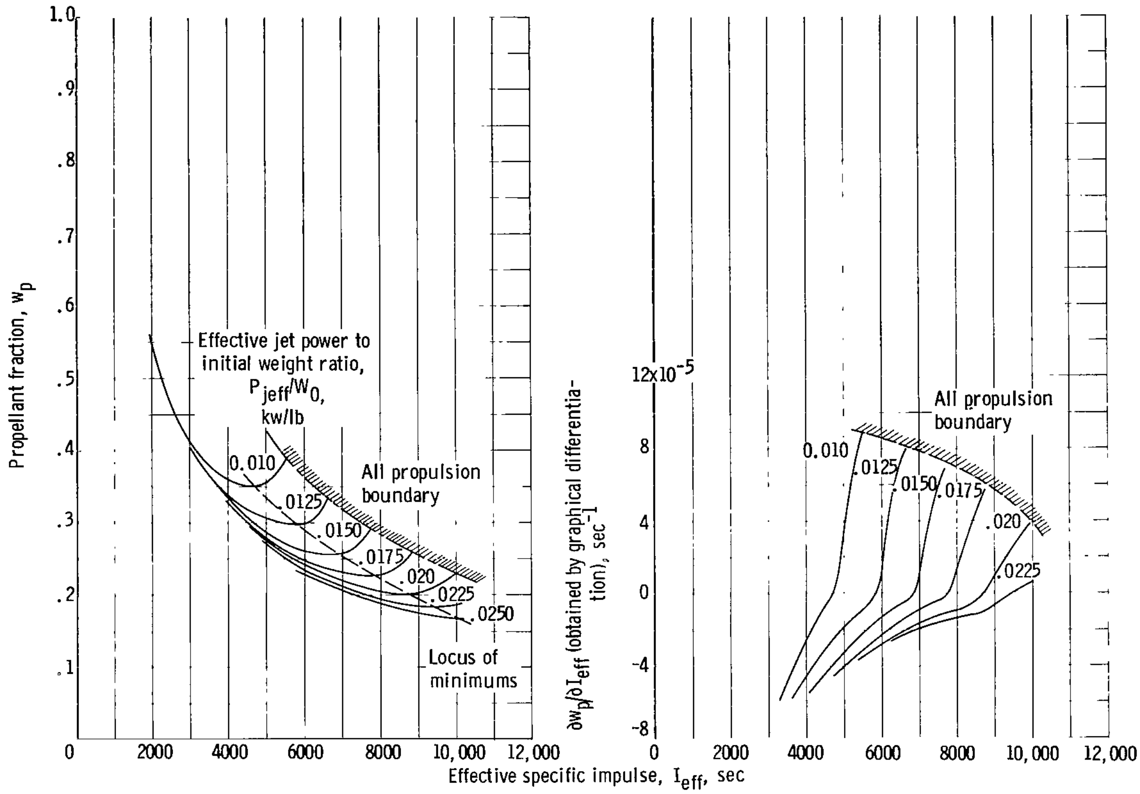


(a) Payload fraction as function of effective specific impulse for $\eta = 1.0$ and $\eta = \eta_{max}(I_{eff})$.

(b) Optimum effective jet power to initial weight ratio as function of effective specific impulse for $\eta = 1.0$ and $\eta = \eta_{max}(I_{eff})$.

Figure 14. - Effects of thrustor efficiency for Mars orbiting probe. Total travel time, 300 days; specific powerplant weight, 10 pounds per kilowatt; structural factor, 0.

The reason for these net effects is best explained with the aid of figure 15, which gives w_p and $\partial w_p/\partial I_{eff}$ as a function of I_{eff}. For $\eta = 1.0$, $\partial w_p/\partial I_{eff}$ must be zero for optimum w_L, and as seen from equation (B16b),



(a) Propellant fraction as function of vehicle performance parameters.

(b) Partial derivative of propellant fraction with respect to effective specific impulse as function of vehicle performance parameters.

Figure 15. - Trajectory results for 300-day Mars orbiting probe.

$\partial w_p/\partial I_{eff}$ must be positive for $\eta = \eta_{max}(I_{eff})$. Figure 15(b) shows that the actual slope $\partial w_p/\partial I_{eff}$ (at constant P_{jeff}/W_0) changes quite rapidly near zero. Therefore, it is possible to satisfy equation (B16b) with almost any efficiency function over a very small range of I_{eff} for $\partial w_p/\partial I_{eff} = 0$. For example, $w_p = 0.257$ and $I_{eff} = 6870$ seconds at $P_{jeff}/W_0 = 0.015$ kilowatt per pound where $\partial w_p/\partial I_{eff} = 0$ (fig. 15). Using the thrustor efficiency assumed in figure 12 results in $\partial w_p/\partial I_{eff} = 1.0\times10^{-5}$ second^{-1}, which gives $w_p = 0.259$ and $I_{eff} = 7000$ seconds. This is an increase of only 130 seconds in I_{eff}. Since this shift in I_{eff} is small at constant P_{jeff}/W_0, the change in propellant required is also small - for the previous example, w_p increases about 1.2 percent of W_0 for the decrease in η from 1.0 to 0.740. Thus, the decrease in payload fraction (5.6 percent of W_0) shown in figure 14 is caused mainly by an increase in powerplant fraction (4.4 percent of W_0) because of an effective increase in α. Since the powerplant fraction will increase over the entire range of P_{jeff}/W_0 and I_{eff} for η less than 1.0, the overall maximum w_L will occur at a lower P_{jeff}/W_0.

With the aforementioned arguments, minimum w_p would serve as a good estimate for the optimum w_p for any similar efficiency function, $\eta = \eta_{max}(I_{eff})$. To estimate payload fractions, only α need be modified by $1/\eta$ for the efficiency function assumed.

A final point to be made about the example concerns the calculation procedure. Normally when the effect of a parameter (e.g., efficiency) is studied, payload fractions are computed and the results plotted to determine the optimums. As shown here, the calculation procedure can be shortened if auxiliary plots of slopes of w_p against P_{jeff}/W_0 and I_{eff} are available. In preparing the results of figure 14 for $\eta = \eta_{max}(I_{eff})$, five data points were calculated by means of the method using slopes. At least three or four times that many calculations are needed to produce the same curve if the optimums are to be determined by plotting. Thus, repeated w_L calculations would definitely warrant the construction of the auxiliary plots of the slopes from the w_p data and the use of the method given previously.

Maximum Payload

The problem of maximum payload is developed here for the 300-day Mars orbiting probe mission. As stated previously, maximum payload and maximum payload fraction are not synonymous. For this reason the conditions necessary for this second type of optimum are developed here. In the example given, it is assumed that the electric powerplant weighs 10 pounds per kilowatt and $k_s = 0$. The same state-of-the-art electron-bombardment thrustors used to illustrate the effect of efficiency are also assumed. Thus, for this case payload is

$$W_L = W_0\left\{1.0 - w_p\left[\eta\left(\frac{\mathscr{P}}{W_0}\right), I_{eff}\right] - \alpha\left(\frac{\mathscr{P}}{W_0}\right)\right\} \tag{B17}$$

where η is considered, as before, to be a function of only I_{eff} and η_u. Therefore, from equation (B17) it is seen that there are four independent variables - η_u, I_{eff}, $\mathscr{P}/W_0$, and W_0. These four variables, however, cannot be

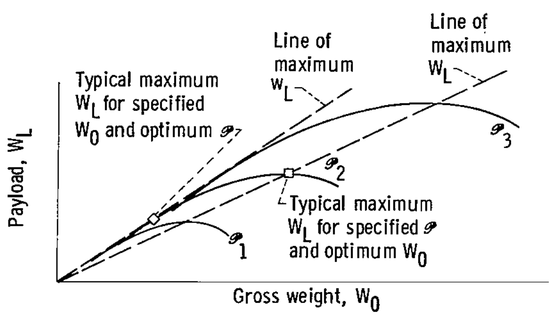


Figure 16. - Typical effect of gross weight and total power on payload.

simultaneously optimized. Hence, two special cases of maximum W_L treated herein are the following:

(1) Specify W_0 and maximize W_L with respect to η_u, I_{eff}, and $\mathscr{P}$.

(2) Specify $\mathscr{P}$ and maximize W_L with respect to η_u, I_{eff}, and W_0.

To clarify the difference between them, these two cases are illustrated in figure 16 where W_L is shown as a function of W_0 for several different values of $\mathscr{P}$.

The first case is identical to the case of maximum payload fraction treated previously. Hence, all the criteria necessary for an optimum must identify the same set of η_u, I_{eff}, and $\mathscr{P}$. Hereinafter, this case will be referred to as maximum w_L. Differentiating equation (B17) for the second case gives

$$dW_L = (1 - w_p)dW_0 - W_0 \frac{\partial w_p}{\partial I_{eff}} dI_{eff} - W_0 \frac{\partial w_p}{\partial\left(\frac{P_{jeff}}{W_0}\right)} d\left(\frac{P_{jeff}}{W_0}\right) \tag{B18}$$

where

$$d\left(\frac{P_{jeff}}{W_0}\right) = \frac{-\eta\mathscr{P}}{W_0^2} dW_0 + \frac{\mathscr{P}}{W_0}\left(\frac{\partial\eta}{\partial I_{eff}} dI_{eff} + \frac{\partial\eta}{\partial\eta_u} d\eta_u\right) \tag{B19}$$

Substituting equation (B19) into equation (B18) and rearranging yield

$$dW_L = -\left\{\mathscr{P}\left[\frac{\partial w_p}{\partial\left(\frac{P_{jeff}}{W_0}\right)}\right]\frac{\partial\eta}{\partial\eta_u}\right\}d\eta_u - \left\{W_0\left(\frac{\partial w_p}{\partial I_{eff}}\right) + \mathscr{P}\left[\frac{\partial w_p}{\partial\left(\frac{P_{jeff}}{W_0}\right)}\right]\frac{\partial\eta}{\partial I_{eff}}\right\}dI_{eff}$$
$$+ \left\{(1 - w_p) + \frac{P_{jeff}}{W_0}\left[\frac{\partial w_p}{\partial\left(\frac{P_{jeff}}{W_0}\right)}\right]\right\}dW_0 \tag{B20}$$

Then for an overall optimum with $\mathscr{P}$ specified, the following conditions must be satisfied:

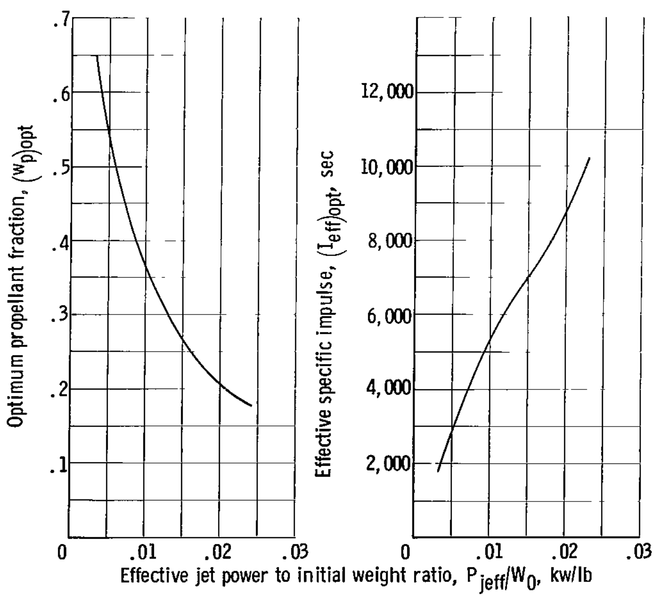
$$\eta_u: \qquad \frac{\partial\eta}{\partial\eta_u} = 0 \tag{B21a}$$

$$I_{eff}: \qquad \frac{\partial w_p}{\partial I_{eff}} = -\frac{1}{\eta}\frac{P_{jeff}}{W_0}\frac{\partial w_p}{\partial\left(\frac{P_{jeff}}{W_0}\right)}\frac{\partial \eta}{\partial I_{eff}} \tag{B21b}$$

$$W_0: \qquad \frac{\partial w_p}{\partial\left(\frac{P_{jeff}}{W_0}\right)} = -\frac{1 - w_p}{\frac{P_{jeff}}{W_0}} \tag{B21c}$$

Several points are to be noted about the criteria for maximum W_L. As expected, equation (B21a) requires maximum overall efficiency. Thus, using the data given in figure 12 (p. 20) satisfies one of the criteria for an optimum. A comparison of these criteria with the case where $\mathscr{P}$ is specified to maximize w_L results in the conclusion that maximum W_L is an off-optimum payload fraction. The final point to be made concerns the optimum vehicle performance parameters for maximum W_L with $\mathscr{P}$ specified. Note that equations (B21) do not contain α. Thus, the optimum P_{jeff}/W_0 and I_{eff} (hence w_p) do not depend on the specific powerplant weight.

From the previous arguments, the results for maximum w_L are readily obtained from the previous section; however, for maximum W_L with $\mathscr{P}$ specified, the calculation procedure used here consists of satisfying equations (B21a) and (B21b) by using figures 12, 13(b), and 15. The result is $(w_p)_{opt}$ and $(I_{eff})_{opt}$ as functions of P_{jeff}/W_0 (fig. 17). Payload calculations were then made for a typical set of parameters, and the maximum W_L determined. This method gave the optimum vehicle parameters for any $\mathscr{P}$ and α with the assumed thrustor efficiency and total travel time.



(a) Optimum propellant fraction as function of effective jet power to initial weight ratio.

(b) Optimum effective specific impulse as function of effective jet power to initial weight ratio.

Figure 17. - Optimum propellant fraction and effective specific impulse for case of maximum payload with specified initial weight and input power for Mars orbiting probe. Total travel time, 300 days; overall efficiency, $\eta = \eta_{max}(I_{eff})$.

Typical results are shown in figure 18 where W_L is plotted against W_0 for $\mathscr{P}$ = 300 kilowatts. On the figure, the case of maximum w_L (data point) gives a payload of 7100 pounds with W_0 = 13,100 pounds. At maximum

W_L, the payload is 11,000 pounds with $W_0 = 27,500$ pounds. Therefore, in this case, 1.55 times the payload can be carried by operating the off-optimum payload fraction; however, this gain is made at the expense of 2.10 times the gross weight in orbit.

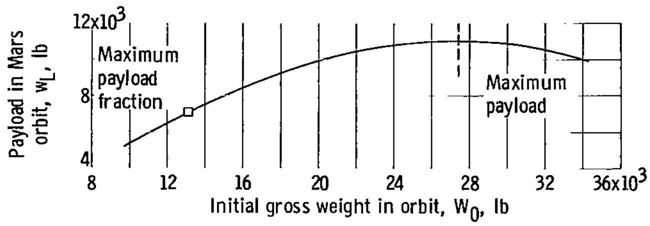


Figure 18. - Maximum payload in Mars orbit as function of initial weight. Total travel time, 300 days; input power to thrustors, 300 kilowatts; specific powerplant weight 10 pounds per kilowatt; structural factor, 0; overall efficiency, $\eta = \eta_{max}(I_{eff})$.

With 27,500-pound booster capability, a better system would consist of two 300-kilowatt powerplants integrated into one spacecraft. From figure 18, the payload for this case would be $2 \times 7400 = 14,800$ pounds - 1.35 times the maximum W_L with the single 300-kilowatt powerplant.

To summarize, the best payload delivered by a spacecraft with a given powerplant and initial gross weight is given directly from a curve such as that shown in figure 18, which satisfies the criteria given by equations (B21a) and (B21b). If the specified W_0 is that value at maximum w_L, then certainly no better payload can ever be obtained; however, if W_0 is that value at maximum W_L for the given powerplant, then the optimum $\mathscr{P}$ should be computed from the optimum vehicle performance parameters at maximum w_L. When this optimum power is more than twice the given power, more payload can be delivered with a cluster of two identical powerplants (assuming that this is possible). This option is available for low-weight powerplants where the P_{jeff}/W_0 at maximum w_L is more than twice the value of P_{jeff}/W_0 at maximum W_L.

At high powerplant specific weights, maximum payload fraction occurs closer to maximum payload. This is true because as α increases, the criteria (eq. (B16a)) approaches equation (B21c), which is not a function of α. As maximum w_L and maximum W_L are the two most important cases treated, the results for the 300-day mission with state-of-the-art electron-bombardment thrustors are summarized in table I.

TABLE I. - SUMMARY OF RESULTS

	Optimum effective specific impulse, $(I_{eff})_{opt}$, sec	Effective jet power to initial weight ratio, P_{jeff}/W_0, kw/lb	Maximum efficiency, η_{max}	Propellant fraction, w_p	Powerplant fraction, w_{pp}	Payload fraction, w_L
Maximum payload fraction[a]	7900	0.0175	0.765	0.231	0.229	0.540
Maximum payload	3400	0.0060	0.550	0.491	0.109	0.400

[a]Powerplant specific weight, 10 lb/kw.

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